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#4 KCooper 1-20-00

Attorney Docket No. 3600.100

IN THE UNITED STATES PATENT AND TRADEMARK OFFICE

In re Reissue Application of	)	Examiner: Not Yet Assigned
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DAVID A. SPEAR ET AL.	)	Group Art Unit: 3745
	:	
Appln. No.: 09/343,736	)	
	:	
Filed: June 30, 1999	)	
	:	
For: SWEPT TURBOMACHINERY BLADE	) :	Application to reissue U.S. Patent No. 5,642,985

Assistant Commissioner for Patents
Box Patent Application
Washington, D.C. 20231

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DECLARATION OF HARRIS D. WEINGOLD
UNDER 37 C.F.R. § 1.132

Sir:

I, Harris D. Weingold, do hereby declare as follows:

1. In 1960 I received a Bachelor of Aeronautical Engineering from New York University. In 1961 I received a Master of Aeronautical Engineering from New York University.

2. From 1965 until June 1999, when I retired, I was employed by the Pratt & Whitney Division of United Technologies Corporation, which on information and belief is the assignee of U.S. Patent No. 5,642,985 to David A. Spear et al., the reissue of which I am informed is sought by the above-identified present application. Throughout my employment with United Technologies Corporation, I have been involved in the analysis, design and production of turbomachinery, and my primary responsibilities

since at least as early as 1974 have been in the analysis of the flow in axial-flow compressors and fans for gas turbine engines, and in research and development of same.

3. I am named as an author on 14 publications and reports in the field of compressors and compressor blades for axial-flow gas turbine engines. A list of my publications is included in my *curriculum vitae*, attached hereto as Exhibit A.

4. I have developed expertise in the analysis of the flow through axial flow turbomachinery, particularly compressors and fans, for gas turbine engines, and in the physics of the interaction of the gas turbine engine working medium gas with the blades and ducts of fans for such gas turbine engines. I am named as an inventor on U.S. Patents No. 4,726,737 and No. 5,088,892.

5. In connection with this declaration I have studied the specification and claims 1-41 of the present application to reissue U.S. Patent 5,642,985 to David A. Spear et al., attached hereto as Exhibit B, and the references listed on the Form PTO-1449 attached hereto as Exhibit C.

6. Claims 1-41 of the present application recite turbomachinery blades for a turbine engine comprising a plurality of such blades arranged so that neighboring blades form passages for a working medium gas. Certain operational conditions produce supersonic velocities in at least some regions of the flow over the blades and through the cascade. While supersonic flow has

more energy (in the form of increased momentum) than subsonic flow, it also creates shocks in the flow. This is explained in the present specification at page 3, lines 30-57; Figures 3, 5 and 7.

7. Before the present invention, those skilled in the art knew that introducing so-called sweep into the leading edge of a turbomachinery blade reduced momentum losses introduced by shocks in the interblade passages, the reason being that a swept blade forms shocks that are oblique to the direction of the fluid flow. An oblique shock creates less momentum loss, so a swept supersonic turbomachinery blade generally is more efficient than an unswept counterpart. However, the configuration of prior art swept blades introduced operational inefficiencies of their own, caused at least in part by multiple shocks and shocks outside of the passages between neighboring blades. Claims 1-41 recite blades with structure that can avoid the inefficiencies and stability problems introduced by prior art swept blades.

8. There are numerous prior art U.S. and foreign patents listed in Exhibit C that disclose turbomachinery blades, some of which cause the working medium gas to attain supersonic velocities in at least some regions of the flow over the blades and through the blade cascade. However, none of them discloses blades with swept configurations like those recited in claims 1-41.

9. Claims 1 and 2 recite blades in which a radially outward region of the blade leading edge is swept such that a section of the blade radially coextensive with an endwall shock extending from a neighboring blade intercepts the endwall shock so that it and a passage shock extending across the flow passage are coincident.

10. It is my opinion that claims 1 and 2 of the present application recite subject matter that would not have been obvious from any of the prior art relating to turbomachinery blades I have studied in connection with this Declaration. None of that prior art would have suggested to one of ordinary skill in this art a blade leading edge swept to intercept an endwall shock so that endwall and passage shocks are coincident.

11. U.S. Patent 5,064,345 ("the Kimball '345 Patent") shows a ventilation-type fan with blades having a leading edge profile similar to the rear swept embodiment shown in Figure 2 of the present application. It discloses a blower-type fan for a heat exchanger or for forced air heating, as explained, for example, at column 1, lines 4-8. However, it contains no teaching that would have been instructive to one of ordinary skill in the art regarding turbomachinery blades for turbine engines.

12. Although the Kimball '345 Patent shows a blade with a swept leading edge, that blade is not suitable for use in a turbine engine, let alone one having blades that rotate fast

enough to produce regions of supersonic flow. It suggests at column 3, lines 22-23, that the fan be manufactured "by conventional plastic molding techniques," which would preclude its rotation at speeds required for use in a turbine engine or that would produce relative supersonic flow over a portion of the blade. Therefore, the Kimball '345 Patent does not disclose a turbomachinery blade for a turbine engine in which shocks form in the flow.

13. The preferred embodiment of the Kimball '345 Patent's fan has an outer band 20, which is intended to improve the strength of the fan. However, providing such a band would not render a fan with the structure disclosed in the Kimball '345 Patent capable of rotation at speeds required for use in a turbine engine or to produce relative supersonic flow over a portion of the blade. The Kimball '345 Patent mentions at column 3, lines 30-31, that its fan need not include a band, but in that event the resulting blade would have a configuration even more clearly incapable structurally of being rotated at speeds producing supersonic flow velocities in at least some regions of the flow over the blades.

14. As a result, one of ordinary skill in the art would not have found the subject matter of claims 1 and 2 obvious from the Kimball '345 Patent.

15. French Patent 2,459,387 ("the French '387 Patent"; citations herein are to the enclosed English translation) shows a

ventilation-type fan with blades having a leading edge profile similar to the forward swept embodiment shown in Figure 6 of the present application. It is representative of other prior art that shows this type of ventilation fan, such as U.S. Patents No. 1,964,525 and No. 4,737,077.

16. Like the Kimball '345 Patent, the configuration of the ventilation fan in the French '387 Patent would preclude its rotation at speeds required for use in a turbine engine or that would produce relative supersonic flow over a portion of the blade. Moreover, the use of bolts to attach the blades to the hub, as seen in Figures 1 and 2, also would preclude~~s~~ their rotation at speeds that would achieve relative supersonic flow in at least some regions of the flow over the blades.

17. As a result, one of ordinary skill in the art would not have found the subject matter of claims 1 and 2 obvious from the French '387 Patent.

18. Puterbaugh et al., "Design of a Rotor Incorporating Meridional Sweep and Circumferential Lean for Shock Loss Attenuation," Contract AFWAL-TR-86-2013, February 1987, Aero Propulsion Laboratories, Air Force Wright Aeronautical Laboratories, Wright-Patterson Air Force Base, Ohio ("the Puterbaugh Report"), reports the results of analytical design studies of the shock losses in swept blades in a turbomachinery rotor. In particular, it reports such studies on a swept rotor blade with increased "circumferential lean," which produced a

blade having ~~with~~ a leading edge with an aerodynamic sweep angle distribution plotted by the lower line in Figure 5 of the Puterbaugh Report, as described below. The Puterbaugh Report uses the term "aerodynamic sweep angle" in generally the same manner as the present application. That is, aerodynamic sweep angle is a term known to those skilled in this art as referring to the angle formed by the blade leading edge and a plane normal to the relative velocity vector of the working medium gas. That is how the term is used in the present application. <u>See</u> the "sweep angle σ" shown in Figure 4 and defined at page 3, lines 20-29, of the present application.

19. The Puterbaugh Report contains no teaching regarding the development of endwall shocks or providing a blade with a leading edge swept to intercept endwall shocks. Therefore, it fails to suggest the blade of claim 1 or 2, with a leading edge swept to intercept an endwall shock so that endwall and passage shocks are coincident.

20. Figure 5 of the Puterbaugh Report contains two plots for the blade discussed in the report. The table in the Puterbaugh Report beginning at the bottom of page 133 and extending to page 134 describes the distribution of the aerodynamic sweep angle along the leading edge ("LE Sweep") and the aerodynamic sweep angle of the passage shock as it intersects the blade suction surface ("Shock Sweep @ Suct Surf"). Comparing these distributions reveals that the magnitude of Shock Sweep @

Suct Surf is greater than that of LE Sweep at every blade radius. These distributions are plotted in Figure 5 as functions of the normalized blade radius, so that it is evident that the upper line in Figure 5 represents Shock Sweep @ Suct Surf and the lower line represents LE Sweep. Accordingly, the upper line does not represent the sweep angle of the blade's leading edge, as recited in the applicants' claims, and one skilled in the art would not have found it to suggest a blade leading edge swept as recited in the applicants' claims.

21. As noted above, the lower line in Figure 5 represents the sweep angle of the leading edge of the Puterbaugh Report's blade. However, that sweep angle begins increasing from a point about 35% of the way from the blade's root end to its tip end, which point is comparable to the beginning of the intermediate region of the blade disclosed in the present application (see, for example, Figure 2). Once it begins to increase at that point, the sweep angle of the leading edge of the Puterbaugh Report's blade increases all the way to the tip. In contrast, a preferred embodiment of the present invention's blade has a leading edge sweep angle that decreases in a tip region, or put another way, has a leading edge with a tip region that is translated in a direction opposite to that of the sweep of the leading edge intermediate region. Such a configuration provides more efficient operation, in one embodiment enabling the blade to intercept an endwall shock so that endwall and passage

shocks are coincident. The Puterbaugh Report does not disclose that blade structure. Accordingly, a blade with the leading edge swept as shown in the lower line of Figure 5 of the Puterbaugh Report also fails to suggest the blade of claim 1 or 2, with a leading edge swept to intercept an endwall shock so that endwall and passage shocks are coincident.

22. Cheatham et al., "Parametric Blade Study," Report No. WRDC-TR-89-2121, November 1989, Aero Propulsion and Power Laboratory, Wright Aeronautical Research & Development Center, Wright-Patterson Air Force Base, Ohio ("the Cheatham Report") listed on the enclosed Form PTO-1449 contains photographs of various turbomachinery blades, having varying degrees of leading edge sweep, tested by the United States Air Force in the 1970's. Figures B18, B19 and B20 are photographs of the blade discussed in the Puterbaugh Report. None of the other blades depicted in the Cheatham Report are believed to be relevant to the present invention. Examination of the front and side views of the Puterbaugh Report's blade (Cheatham Report Rotor 7), and of the other rotors depicted in the Cheatham Report, reveals no feature suggesting the particular leading edge sweep distribution that is a salient feature of the present invention.

23. There are other prior art patents listed on Exhibit C that show various blade shapes used in different applications, such as ventilation fans (U.S. Patents No. 2,154,313, No. 3,416,725 and No. 4,737,007, and Soviet Patent SU

1,528,965), a fluid pump (U.S. Patent No. 3,444,817) a mixing impeller for blending liquids (U.S. Patent No. 5,112,192), aircraft prop fans (U.S. Patents No. 4,370,097 and No. 4,358,246) and helicopter rotor blades (U.S. Patent No. 5,584,661). None of them disclose turbomachinery blades like those recited in claims 1-41, or solve (or even recognize the existence of) the problems capable of solution by the blades recited in such claims. As a result, one of ordinary skill in the art would not have found the invention of claims 1 and 2 obvious from any of those references. For example, U.S. Patent No. 5,584,661 discloses a swept helicopter rotor blade in which shocks form in the blade tip region. However, the sweep profile of the blade is unlike that of the turbomachinery blade of the present invention and is incapable of suggesting to one of ordinary skill in the art a turbomachinery blade with a leading edge swept to intercept an endwall shock or so that endwall and passage shocks are coincident.

24. Claim 4 is directed to a swept turbomachinery blade for a gas turbine engine fan comprising a plurality of such blades in which neighboring blades form passages for a working medium gas. The blade is configured to enable the cascade to rotate at speeds that provide supersonic flow velocities in at least a portion of the interblade passages, causing a shock adjacent the inner wall of a case that forms the outer boundary for the working medium gas flowing through the cascade. The

blade leading edge has a rear swept intermediate region and a tip region that is translated forward (as shown in the embodiment depicted in Figure 2 of the present application) to provide a sweep angle that causes the blade to intercepts the shock. Claim 4 therefore would not have been obvious to one of ordinary skill in the art for at least the same reasons discussed above in connection with claims 1 and 2. That is, the prior art I have studied in connection with this Declaration, including that relating to turbomachinery blades, does not contain any suggestion of a supersonic turbomachinery blade with an intermediate region that is swept rearward and a tip region that is translated forward to provide a sweep angle that causes the blade to intercept a shock formed at the wall of a case.

25. Claim 10 recites a gas turbine engine fan blade also comprising a plurality of such blades in which neighboring blades form passages for a working medium gas. As in claim 4, the blade is configured to enable the cascade to rotate at speeds that provide supersonic flow velocities in at least a portion of the interblade passages. The blade of claim 10 has a leading edge with a swept intermediate region and a tip region that begins at a radially outward boundary of the intermediate region and extends to a tip end of the blade. Throughout the tip region the leading edge sweep angle is less than it is at the beginning of the tip region (that is, at the intermediate region/tip region boundary). Claim 20 recites turbomachinery including a blade

with this feature. The subject matter of each of claims 10 and 20 would not have been obvious to one of ordinary skill in the art for reasons discussed above in connection with claims 1, 2 and 4. That is, none of the prior art listed on the enclosed Form PTO-1449, including that relating to turbomachinery blades, contains any suggestion of a supersonic turbomachinery blade with a tip region throughout which the leading edge sweep angle is less than it is at the tip region's boundary with an intermediate region.

26. As noted above, Figure 5 of the Puterbaugh Report discloses a blade with a rearwardly swept leading edge. (Claims 10 and 20 relate to blades with either a rearward or forward swept intermediate region.) As I stated above, the lower line in Figure 5 represents the sweep angle of the blade's leading edge, which clearly does not include a region extending to the tip of the blade in which the sweep angle throughout such region is less than the sweep angle at the radially inner boundary of the region. This is more than simply a minor difference in blade structure, since it is the recited leading edge profile that provides the recited blade with the superior aerodynamic properties described in the present application.

27. Claims 18, 27 and 30 recite gas turbine engine blades with a rearward swept intermediate or middle region and a tip region that is translated forward from the intermediate region's outer boundary, at which the tip region begins. Claim

36 recites a gas turbine engine blade with a forward swept middle region and a tip region that is translated rearward from the middle region's outer boundary. The subject matter of each of claims 18, 27, 30 and 36 is patentable for reasons discussed in detail above in connection with the other claims in the present application. That is, none of the prior art listed on the enclosed Form PTO-1449 discloses or suggests a supersonic turbomachinery blade with a leading edge that is swept in one direction (rearward or forward) and a tip region that is translated in the other direction from its boundary with the intermediate region.

I hereby declare that all statements made herein of my own knowledge are true and that all statements made on information and belief are believed to be true; and further that the statements were made with the knowledge that willful false statements and the like so made are punishable by fine or imprisonment, or both, under § 1001 of Title XVIII of United States Code, and that such willful false statements made jeopardize the validity of this application or any patent issued thereon.

Date: 12/27/99

Harris D. Weingold

Harris D. Weingold

Exhibit A

Harris D. Weingold
51 Vardon Road
West Hartford, CT 06117

Date of Birth - 1/19/1940 | Tel. 860-233-4075
Married | e-mail Turbowhiz@aol.com

Employment

Current Independent Consultant - Turbomachinery

1965 - 1999 Pratt & Whitney Division, United Technologies Corporation
(Retired from Pratt & Whitney June 30, 1999)

Scientific Analysis Group 1965-1974

Developed viscous boundary layer programs for use in aircraft nozzle design, under AF contract.
Developed rocket nozzle boundary layer code selected as industry standard by ICRPG. Developed choked venturi viscous discharge coefficient prediction, used in P&W large test stand flowmetering system development.
Developed 2D/3D/Axisymmetric incompressible potential flow panel codes. Demonstrated first capability at P&W for analysis of complex flowfields: thick cascades, slotted cascades, multibody flows, fan EGV/strut interaction, nacelle ground plane interaction/vortex ingestion, compressor and turbine intermediate cases, bifurcated exhaust plenums, etc.

Compressor Technology and Research Group 1974-1979

Planning and budgetary responsibility for compressor technology programs.
Directed test programs in high speed and low speed cascade tunnels, and 3-stage compressor research facilities.
Directed and participated in development of analytical methods to design and predict performance of compressible compressor cascade sections. Experimentally demonstrated accuracy of program, and eliminated dependence on correlations and extrapolations of cascade test data in design process. Discontinued cascade testing.
Directed and participated in development of supercritical cascade sections and expanded program to cover all compressor airfoil sections under the Controlled Diffusion Airfoil redesignation. Experimentally demonstrated the superiority of CDAs relative to standard airfoil sections. CDAs are now used in 95% of all recent P&W compressor rows and are credited with 1% improvement in compressor efficiency.

Compressor Analysis and Technology Development Group 1979-1993

Planning and budgetary responsibility for compressor technology. Responsible for the coordination of the Pratt & Whitney IR&D Technical plan Compressor Sections, and presentation at On-Site Reviews as well as periodic technology review visits to NASA and USAF.
Directed and participated in development & application of advanced analytical methods for fan and compressor design: 3D Euler, Quasi-3D design methods, viscous cascade analyses, & 3D viscous. Won 1988 ASME Gas Turbine Award for 'Prediction of Compressor Cascade Performance Using a Navier-Stokes Technique' with R.L. Davis and D.E. Hobbs, ASME Journal of Turbomachinery, Vol. 110, October 1988.
Pioneered use of 3D analytical methods to depart from traditional design paradigms to achieve performance breakthroughs. Used 3D Euler and later 3D Navier-Stokes analyses to attack endwall loss problem. Demonstrated the fallacy of the traditional endwall boundary layer concept and demonstrated the benefits and necessity of full span design of compressor airfoils. Developed 3D airfoil shapes which substantially reduced or eliminated endwall loss in compressor stators. Bowed Stators (patented) were shown to improve compressor efficiency by 1.1% to 1.9% in 3-stage rig testing and are credited with

contributing to a 2.3% efficiency improvement when applied to the PW4098 HPC. Now incorporated in the F119 HPC as well.
Applied a 3D Euler analysis to demonstrate the potential to substantially reduce shock loss in transonic fan blades through the use of sweep (patented), and developed stacking procedures to avoid excessive bending stresses while achieving desired degree of sweep.

Compressor Methods Group & Aero-X Group 1994-1999
Developed understanding of the physics of rotor tip wall stall and described the role of compressor design variables in contributing to a propensity toward stability sensitivity to tip clearance in recent designs. Theory explains the effects of stator cutbacks towards alleviation of this problem.

Education

1960 — Bachelor of Aeronautical Engineering, Summa Cum Laude
New York University
New York State Regents Scholarship and Bendix Aviation Corp. Scholarship

1961 — Master of Aeronautical Engineering
New York University
National Science Foundation Cooperative Graduate Fellowship

to 1965 — Continued postgraduate work in Aeronautical Engineering at New York University, supported by National Science Foundation Cooperative Graduate Fellowships

Academic Honors

Tau Beta Pi - National engineering honorary
Sigma Gamma Tau - Aeronautical Engineering Honorary
Tau Beta Pi Outstanding Freshman Award -1957
Alexander Klemin Award in Aeronautical Engineering, NYU 1960
William Remington Bryant Award in Engineering Mechanics, NYU 1960

Continuing Education

1975 — ASME/AMES Fluid Dynamics of Turbomachinery. Iowa State University

1986 — Zenger-Miller Leadership Development Program

1986-1987 — TQM Training

Professional Affiliations

AIAA, Associate Fellow
Chairman of Connecticut Section, AIAA, 1978-1979
Officer and Council Member, Conn. Section, 1974-1980, 1987-1994

ASME , Member
Turbomachinery Committee, 1995 - present

Professional Honors

1988 ASME Gas Turbine Award

Exhibit A

Publications

'Flow Through Cascades of Slotted Compressor Blades,' with A.A. Mikolajczak and J.P. Nikkanen, ASME Journal of engineering for Power, Vol. 92, Ser. A, No. 1, January 1970

'Vibratory Forcing Functions Produced by Nonuniform Cascades,' with T.J. Barber, ASME Journal of Engineering for Power, Vol. 100, No. 1, January 1978

'Development of Controlled Diffusion Airfoils for Multistage Compressor Applications,' with D.E. Hobbs, ASME Journal of Engineering for Gas turbines and Power, Vol. 106, No, 2, April 1984

'Experimental Investigation of a Simulated Compressor Airfoil Trailing Edge Flowfield,' with R.W. Paterson, AIAA Journal, Vol. 21, No. 5, May 1985

'The Use of Surface Static Pressure Data as a Diagnostic Tool in Multistage Compressor Development,' with R.F. Behlke, ASME Journal of Turbomachinery, Vol. 109, January 1987

'Prediction of Compressor Cascade Performance Using a Navier Stokes Technique,' with R.L. Davis and D.E. Hobbs, ASME Journal of Turbomachinery, Vol. 110, October 1988
*** Won ASME Gas Turbine Award as Best Gas Turbine Paper of 1988**

'Experimental Investigation of Loading Effects on Simulated Compressor Airfoil Trailing-Edge Flowfields,' with D.C. McCormick and R.W. Paterson, AIAA Journal of Propulsion and Power, Vol. 7, No. 2, March-April 1991

'Application of Sweep to Improve The Efficiency of a Transonic Fan, Part I - Design,' with R.J. Neubert and D.E. Hobbs, AIAA Journal of Propulsion and Power, Vol. 11, No. 1, January 1995

'Reduction of Compressor Stator Endwall Losses Through the Use of Bowed Stators,' with R.J. Neubert, R.F. Behlke, and G.E. Potter, 95-GT-380, 1995 ASME Gas Turbine Conference, Houston, TX, June 1995

'A Navier-Stokes Analysis of the Effect of Tip Clearance on Compressor Stall Margin,' with R.P. Dring and W.D. Sprout, 1995 ASME Gas Turbine Conference, Houston, TX, June 1995

'Bowed Stators - An Example of CFD Applied to Improve the Efficiency of Multistage Compressors,' with R.J. Neubert, R.F. Behlke, and G.E. Potter, ASME Journal of Turbomachinery, April 1997

Contract Reports

" TBL 'ICRPG Turbulent Boundary Layer Nozzle Analysis Computer Program'," published by the Interagency Chemical Rocket Propulsion Group

'Supercritical Airfoil Technology Program - Experimental Investigation of a Simulated Compressor Airfoil Trailing Edge Flowfield,' with R.W. Paterson, PWA Report FR-15859, October 1982

'Energy Efficient Engine Hollow Fan Blade Technology, Volume III - Swept Fan Feasibility Study,' with B.C. Chisolm, and D.J. Dubiel, NASA CR 182223, August 1989

Patents

U.S. Patent 4,726,737, 'Reduced Loss Swept Supersonic Fan Blade'

U.S. Patent 5,088,892, ''Bowed Airfoil for the Compression Section of a Rotary Machine'

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I hereby certify that this paper or fee is being deposited with the United States Postal Service "Express Mail Post Office to Addressee." service under 37 CFR 1.10 on the date indicated above and is addressed to the Commissioner of Patents and Trademarks Washington DC 20231
Karen Malatesta
KAREN MALATESTA

EXHIBIT B

SWEPT TURBOMACHINERY BLADE

TECHNICAL FIELD

This invention relates to turbomachinery blades, and particularly to blades whose airfoils are swept to minimize the adverse effects of supersonic flow of a working medium over the airfoil surfaces.

BACKGROUND OF THE INVENTION

Gas turbine engines employ cascades of blades to exchange energy with a compressible working medium gas that flows axially through the engine. Each blade in the cascade has an attachment which engages a slot in a rotatable hub so that the blades extend radially outward from the hub. Each blade has a radially extending airfoil, and each airfoil cooperates with the airfoils of the neighboring blades to define a series of interblade flow passages through the cascade. The radially outer boundary of the flow passages is formed by a case which circumscribes the airfoil tips. The radially inner boundary of the passages is formed by abutting platforms which extend circumferentially from each blade.

During engine operation the hub, and therefore the blades attached thereto, rotate about a longitudinally extending rotational axis. The velocity of the working medium relative to the blades increases with increasing radius. Accordingly, it is not uncommon for the airfoil leading edges to be swept forward or swept back to mitigate the adverse aerodynamic effects associated with the compressibility of the working medium at high velocities.

One disadvantage of a swept blade results from pressure waves which extend along the span of each airfoil suction surface and reflect off the surrounding case. Because the airfoil is swept, both the incident waves and the reflected waves are oblique to the case. The reflected waves interact with the incident waves and coalesce into a planar aerodynamic shock which extends across the interblade flow channel between neighboring airfoils. These "endwall shocks" extend radially inward a limited distance from the case. In addition, the compressibility of the working medium causes a passage shock, which is unrelated to the above described endwall shock, to extend across the passage from the leading edge of each blade to the suction surface of the adjacent blade. As a result, the working medium gas flowing into the channels encounters multiple shocks and experiences unrecoverable losses in velocity and total pressure, both of which degrade the engine's efficiency. What is needed is a turbomachinery blade whose airfoil is swept to mitigate the effects of working medium compressibility while also avoiding the adverse influences of multiple shocks.

DISCLOSURE OF THE INVENTION

It is therefore an object of the invention to minimize the aerodynamic losses and efficiency degradation associated with endwall shocks by limiting the number of shocks in each interblade passage.

According to the invention, a blade for a blade cascade has an airfoil which is swept over at least a portion of its span, and the section of the airfoil radially coextensive with the endwall shock intercepts the endwall shock extending from the neighboring airfoil so that the endwall shock and the passage shock are coincident.

In one embodiment the axially forwardmost extremity of the airfoil's leading edge defines an inner transition point located at an inner transition radius radially inward of the

airfoil tip. An outer transition point is located at an outer transition radius radially intermediate the inner transition radius and the airfoil tip. The outer transition radius and the tip bound a blade tip region while the inner and outer transition radii bound an intermediate region. The leading edge is swept at a first sweep angle in the intermediate region and is swept at a second sweep angle over at least a portion of the tip region. The first sweep angle is generally nondecreasing with increasing radius and the second sweep angle is generally non-increasing with increasing radius.

The invention has the advantage of limiting the number of shocks in each interblade passage so that engine efficiency is maximized.

BRIEF DESCRIPTION OF THE DRAWINGS

FIG. 1 is a cross sectional side elevation of the fan section of a gas turbine engine showing a swept back fan blade according to the present invention.

FIG. 2 is an enlarged view of the blade of FIG. 1 including an alternative leading edge profile shown by dotted lines and a prior art blade shown in phantom.

FIG. 3 is a developed view taken along the line 3—3 of FIG. 2 illustrating the tips of four blades of the present invention along with four prior art blades shown in phantom.

FIG. 4 is a schematic perspective view of an airfoil fragment illustrating the definition of sweep angle.

FIG. 5 is a developed view similar to FIG. 3 illustrating an alternative embodiment of the invention and showing prior art blades in phantom.

FIG. 6 is a cross sectional side elevation of the fan section of a gas turbine engine showing a forward swept fan blade according to the present invention and showing a prior art fan blade in phantom.

FIG. 7 is a developed view taken along the line 7—7 of FIG. 6 illustrating the tips of four blades of the present invention along with four prior art blades shown in phantom.

BEST MODE FOR CARRYING OUT THE INVENTION

Referring to FIGS. 1–3, the forward end of a gas turbine engine includes a fan section **10** having a cascade of fan blades **12**. Each blade has an attachment **14** for attaching the blade to a disk or hub **16** which is rotatable about a longitudinally extending rotational axis **18**. Each blade also has a circumferentially extending platform **20** radially outward of the attachment. When installed in an engine, the platforms of neighboring blades in the cascade abut each other to form the cascade's inner flowpath boundary. An airfoil **22** extending radially outward from each platform has a root **24**, a tip **26**, a leading edge **28**, a trailing edge **30**, a pressure surface **32** and a suction surface **34**. The axially forwardmost extremity of the leading edge defines an inner transition point **40** at an inner transition radius r_t-inner, radially inward of the tip. The blade cascade is circumscribed by a case **42** which forms the cascade's outer flowpath boundary. The case includes a rubstrip **46** which partially abrades away in the event that a rotating blade contacts the case during engine operation. A working medium fluid such as air **48** is pressurized as it flows axially through interblade passages **50** between neighboring airfoils.

The hub **16** is attached to a shaft **52**. During engine operation, a turbine (not shown) rotates the shaft, and therefore the hub and the blades, about the axis **18** in direction R. Each blade, therefore, has a leading neighbor

which precedes it and a trailing neighbor which follows it during rotation of the blades about the rotational axis.

The axial velocity V_x (FIG. 3) of the working medium is substantially constant across the radius of the flowpath. However the linear velocity U of a rotating airfoil increases with increasing radius. Accordingly, the relative velocity V_r of the working medium at the airfoil leading edge increases with increasing radius, and at high enough rotational speeds, the airfoil experiences supersonic working medium flow velocities in the vicinity of its tip. Supersonic flow over an airfoil, while beneficial for maximizing the pressurization of the working medium, has the undesirable effect of reducing fan efficiency by introducing losses in the working medium's velocity and total pressure. Therefore, it is typical to sweep the airfoil's leading edge over at least a portion of the blade span so that the working medium velocity component in the chordwise direction (perpendicular to the leading edge) is subsonic. Since the relative velocity V_r increases with increasing radius, the sweep angle typically increases with increasing radius as well. As shown in FIG. 4, the sweep angle σ at any arbitrary radius is the acute angle between a line **54** tangent to the leading edge **28** of the airfoil **22** and a plane **56** perpendicular to the relative velocity vector V_r. The sweep angle is measured in plane **58** which contains both the relative velocity vector and the tangent line and is perpendicular to plane **56**. In conformance with this definition sweep angles σ_1 and σ_2, referred to hereinafter and illustrated in FIGS. 2, 3 and 6 are shown as projections of the actual sweep angle onto the plane of the illustrations.

Sweeping the blade leading edge, while useful for minimizing the adverse effects of supersonic working medium velocity, has the undesirable side effect of creating an endwall reflection shock. The flow of the working medium over the blade suction surface generates pressure waves **60** (shown only in FIG. 1) which extend along the span of the blade and reflect off the case. The reflected waves **62** and the incident waves **60** coalesce in the vicinity of the case to form an endwall shock **64** across each interblade passage. The endwall shock extends radially inward a limited distance, d, from the case. As best seen in the prior art (phantom) illustration of FIG. 3, each endwall shock is also oblique to a plane **67** perpendicular to the rotational axis so that the shock extends axially and circumferentially. In principle, an endwall shock can extend across multiple interblade passages and affect the working medium entering those passages. In practice, expansion waves (as illustrated by the representative waves **68**) propagate axially forward from each airfoil and weaken the endwall shock from the airfoil's leading neighbor so that each endwall shock usually affects only the passage where the endwall shock originated. In addition, the supersonic character of the flow causes passage shocks **66** to extend across the passages. The passage shocks, which are unrelated to endwall reflections, extend from the leading edge of each blade to the suction surface of the blade's leading neighbor. Thus, the working medium is subjected to the aerodynamic losses of multiple shocks with a corresponding degradation of engine efficiency.

The endwall shock can be eliminated by making the case wall perpendicular to the incident expansion waves so that the incident waves coincide with their reflections. However other design considerations, such as constraints on the flowpath area and limitations on the case construction, may make this option unattractive or unavailable. In circumstances where the endwall shock cannot be eliminated, it is desirable for the endwall shock to coincide with the passage shock since the aerodynamic penalty of coincident shocks is less than that of multiple individual shocks.

According to the present invention, coincidence of the endwall shock and the passage shock is achieved by uniquely shaping the airfoil so that the airfoil intercepts the endwall shock extending from the airfoil's leading neighbor and results in coincidence between the endwall shock and the passage shock.

A swept back airfoil according to the present invention has a leading edge 28, a trailing edge 30, a root 24 and a tip 26 located at a tip radius r_{tip}. An inner transition point 40 located at an inner transition radius r_r-inner is the axially forwardmost point on the leading edge. The leading edge of the airfoil is swept back by a radially varying first sweep angle σ_1 in an intermediate region 70 of the airfoil (in FIG. 2 plane 56 appears as the line defined by the plane's intersection with the plane of the illustration and in FIG. 3 the tangent line 54 appears as the point where the tangent line penetrates the plane of the Figure). The intermediate region 70 is the region radially bounded by the inner transition radius r_r-inner and the outer transition radius r_r-outer. The first sweep angle, as is customary in the art, is nondecreasing with increasing radius, i.e. the sweep angle increases, or at least does not decrease, with increasing radius.

The leading edge 28 of the airfoil is also swept back by a radially varying second sweep angle σ_2 in a tip region 74 of the airfoil. The tip region is radially bounded by the outer transition radius r_r-outer and a tip radius r_{tip}. The second sweep angle is nonincreasing (decreases, or at least does not increase) with increasing radius. This is in sharp contrast to the prior art airfoil 22' whose sweep angle increases with increasing radius radially outward of the inner transition radius.

The beneficial effect of the invention is appreciated primarily by reference to FIG. 3 which compares the invention (and the associated endwall and passage shocks) to a prior art blade (and its associated shocks) shown in phantom. Referring first to the prior art illustration in phantom, the endwall shock 64 originates as a result of the pressure waves 60 (FIG. 1) extending along the suction surface of each blade. Each endwall shock is oblique to a plane 67 perpendicular to the rotational axis, and extends across the inter-blade passage of origin. The passage shock 66 also extends across the flow passage from the leading edge of a blade to the suction surface of the blade's leading neighbor. The working medium entering the passages is therefore adversely influenced by multiple shocks. By contrast, the nonincreasing character of the second sweep angle of a swept back airfoil 22 according to the invention causes a portion of the airfoil leading edge to be far enough forward (upstream) in the working medium flow that the section of the airfoil radially coextensive with the endwall shock extending from the airfoil's leading neighbor intercepts the endwall shock 64 (the unique sweep of the airfoil does not appreciably affect the location or orientation of the endwall shock; the phantom endwall shock associated with the prior art blade is illustrated slightly upstream of the endwall shock for the airfoil of the invention for illustrative clarity). In addition, the passage shock 66 (which remains attached to the airfoil leading edge and therefore is translated forward along with the leading edge) is brought into coincidence with the endwall shock so that the working medium does not encounter multiple shocks.

The embodiment of FIGS. 2 and 3 illustrates a blade whose leading edge, in comparison to the leading edge of a conventional blade, has been translated axially forward parallel to the rotational axis (the corresponding translation of the trailing edge is an illustrative convenience—the

location of the trailing edge is not embraced by the invention). However the invention contemplates any blade whose airfoil intercepts the endwall shock to bring the passage shock into coincidence with the endwall shock. For example, FIG. 5 illustrates an embodiment where a section of the tip region is displaced circumferentially (relative to the prior art blade) so that the blade intercepts the endwall shock 64 and brings it into coincidence with the passage shock 66. As with the embodiment of FIG. 3. the displaced section extends radially inward far enough to intercept the endwall shock over its entire radial extent and brings it into coincidence with the passage shock 66. This embodiment functions as effectively as the embodiment of FIG. 3 in terms of bringing the passage shock into coincidence with the endwall shock. However it suffers from the disadvantage that the airfoil tip is curled in the direction of rotation R. In the event that the blade tip contacts the rubstrip 46 during engine operation, the curled blade tip will gouge rather than abrade the rubstrip necessitating its replacement. Other alternative embodiments may also suffer from this or other disadvantages.

The invention's beneficial effects also apply to a blade having a forward swept airfoil. Referring to FIG. 6 and 7, a forward swept airfoil 122 according to the present invention has a leading edge 128, a trailing edge 130, a root 124 and a tip 126 located at a tip radius r_{tip}. An inner transition point 140 located at an inner transition radius r_r-inner is the axially aftmost point on the leading edge. The leading edge of the airfoil is swept forward by a radially varying first sweep angle σ_1 in an intermediate region 70 of the airfoil. The intermediate region is radially bounded by the inner transition radius r_r-inner and the outer transition radius r_r-outer. The first sweep angle [τ_1] σ_1 is nondecreasing with increasing radius, i.e. the sweep angle increases, or at least does not decrease, with increasing radius.

The leading edge 128 of the airfoil is also swept forward by a radially varying second sweep angle σ_2 in a tip region 74 of the airfoil. The tip region is radially bounded by the outer transition radius r_r-outer and the tip radius r_{tip}. The second sweep angle is nonincreasing (decreases, or at least does not increase) with increasing radius. This is in sharp contrast to the prior art airfoil 122' whose sweep angle increases with increasing radius radially outward of the inner transition radius.

In the forward swept embodiment of the invention, as in the swept back embodiment, the nonincreasing sweep angle σ_2 in the tip region 74 causes the endwall shock 64 to be coincident with the passage shock 66 for reducing the aerodynamic losses as discussed previously. This is in contrast to the prior art blade, shown in phantom where the endwall shock and the passage shock are distinct and therefore impose multiple aerodynamic losses on the working medium.

In the swept back embodiment of FIG. 2, the inner transition point is the axially forwardmost point on the leading edge. The leading edge is swept back at radii greater than the inner transition radius. The character of the leading edge sweep inward of the inner transition radius is not embraced by the invention. In the forward swept embodiment of FIG. 6, the inner transition point is the axially aftmost point on the leading edge. The leading edge is swept forward at radii greater than the inner transition radius. As with the swept back embodiment, the character of the leading edge sweep inward of the inner transition radius is not embraced by the invention. In both the forward swept

and back swept embodiments. the inner transition point is illustrated as being radially outward of the airfoil root. However the invention also comprehends a blade whose inner transition point (axially forwardmost point for the swept back embodiment and axially aftmost point for the forward swept embodiment) is radially coincident with the leading edge of the root. This is shown, for example, by the dotted leading edge 28" of FIG. 2.

The invention has been presented in the context of a fan blade for a gas turbine engine, however, the invention's applicability extends to any turbomachinery airfoil wherein flow passages between neighboring airfoils are subjected to multiple shocks.

We claim:

1. A turbomachinery blade for a turbine engine having a cascade of blades rotatable about a rotational axis so that each blade in the cascade has a leading neighbor and a trailing neighbor, and each blade cooperates with its neighbors to define flow passages for a working medium gas, the blade cascade being circumscribed by a case and under some operational conditions an endwall shock extends a limited distance radially inward from the case and also extends axially and circumferentially across the flow passages, and a passage shock also extends across the flow passages, the turbomachinery blade including an airfoil having a leading edge, a trailing edge, a root, a tip and an inner transition point located at an inner transition radius radially inward of the tip, the blade characterized in that at least a portion of the leading edge radially outward of the inner transition point is swept and a section of the airfoil radially coextensive with the endwall shock extending from the leading neighbor intercepts the endwall shock so that the endwall shock and the passage shock are coincident.

2. A turbomachinery blade for a turbine engine having a cascade of blades rotatable about a rotational axis so that each blade in the cascade has a leading neighbor and a trailing neighbor, and each blade cooperates with its neighbors to define flow passages for a working medium gas, the blade cascade being circumscribed by a case and under some operational conditions an endwall shock extends a limited distance radially inward from the case and also extends axially and circumferentially across the flow passages and a passage shock also extends across the flow passages, the turbomachinery blade including an airfoil having a leading edge, a trailing edge, a root, a tip located at a tip radius, an inner transition point located at an inner transition radius radially inward of the tip, and an outer transition point at an outer transition radius radially intermediate the inner transition radius and the tip radius, the blade having a tip region bounded by the outer transition radius and the tip radius, and an intermediate region bounded by the inner transition radius and the outer transition radius, the blade characterized in that the leading edge is swept in the intermediate region at a first sweep angle which is generally nondecreasing with increasing radius, and the leading edge is swept over at least a portion of the tip region at a second sweep angle which is generally nonincreasing with increasing radius so that the section of the airfoil radially coextensive with the endwall shock extending from the leading neighbor intercepts the endwall shock so that the endwall shock and the passage shock are coincident.

3. The turbomachinery blade of claim 1 or 2 characterized in that the inner transition radius is coincident with the root at the leading edge of the blade.

4. A turbomachinery blade for a gas turbine engine fan comprising a plurality of blades mounted for rotation about a fan axis with neighboring blades forming passages for a working medium gas, wherein:

the blade has a configuration enabling the fan to rotate at speeds providing supersonic flow velocities in at least a portion of each passage causing the formation of a shock in the gas adjacent an inner wall of a case forming an outer boundary for the working medium gas flowing through the passages;

the blade has a leading edge with an intermediate region and a tip region outward of the intermediate region and extending to a tip end of the blade, the intermediate region being swept rearward at a sweep angle that does not decrease; and

the tip region is translated forward to provide a sweep angle that causes the blade to intercept the shock.

5. The turbomachinery blade of claim 4, wherein the tip region begins at an outward boundary of the intermediate region and throughout the tip region the sweep angle is less than the sweep angle at the outward boundary of the intermediate region.

6. The turbomachinery blade of claim 5, wherein the sweep angle decreases throughout the tip region.

7. The turbomachinery blade of claim 6, wherein the sweep angle increases throughout the intermediate region.

8. The turbomachinery blade of any one of claims 4 to 7, wherein an inward boundary of the intermediate region is coincident with a root end of the blade.

9. The turbomachinery blade of any one of claims 4 to 7, wherein the leading edge of the blade has an inner region beginning at a root end of the blade and extending to an inward boundary of the intermediate region, the inner region being swept forward.

10. A blade for a gas turbine engine fan comprising a plurality of blades mounted for rotation within a case circumscribing the blades and forming an outer boundary for a working medium gas flowing through passages formed by neighboring blades, wherein:

the blade has a configuration enabling the fan to rotate at speeds providing supersonic flow velocities in at least a portion of each passage;

the blade has a leading edge with an intermediate region and a tip region beginning at an outward boundary of the intermediate region and extending to a tip end of the blade, the intermediate region having a sweep angle that does not decrease from the beginning to the outward boundary of the intermediate region; and

throughout the tip region the sweep angle is less than the sweep angle at the outward boundary of the intermediate region.

11. The blade of claim 10, wherein the intermediate region is swept rearward and the tip region is translated forward.

12. The blade of claim 10, wherein the intermediate region is swept forward and the tip region is translated rearward.

13. The blade of claim 10, wherein the tip region sweep angle decreases throughout the tip region.

14. The blade of claim 13, wherein the intermediate region sweep angle increases throughout the intermediate region.

15. The blade of any one of claims 10 to 14, wherein an inward boundary of the intermediate region is coincident with a root end of the blade.

16. The blade of claim 10, wherein:

the intermediate region is swept rearward and the tip region is translated forward; and

the leading edge of the blade has an inner region beginning at a root end of the blade and extending to an inward boundary of the intermediate region, the inner region being swept forward.

17. The blade of claim 16, wherein:

the intermediate region sweep angle increases throughout the intermediate region; and

the tip region sweep angle decreases throughout the tip region.

18. A blade for a gas turbine engine fan comprising a plurality of blades mounted for rotation within a case circumscribing the blades and forming an outer boundary for a working medium gas flowing through passages formed by neighboring blades, wherein:

the blade has a configuration enabling the fan to rotate at speeds providing supersonic flow velocities in at least a portion of each passage;

the blade has a leading edge with an intermediate region and a tip region beginning at an outward boundary of the intermediate region and extending to a tip end of the blade, the intermediate region being swept rearward at a sweep angle that does not decrease from the beginning to the outward boundary of the intermediate region; and

the tip region is translated forward from the outward boundary of the rearwardly swept intermediate region.

19. The blade of claim 18, wherein the tip region maintains a rearward sweep throughout the tip region.

20. Turbomachinery for a gas turbine engine, comprising a plurality of blades mounted for rotation within a case circumscribing the blades and forming an outer boundary for a working medium gas flowing through passages formed by neighboring blades, wherein:

each blade has a configuration enabling the turbomachinery to rotate at speeds providing supersonic working medium gas velocities at least in the vicinity of the passages proximate to the case;

each blade has a leading edge with a swept intermediate region and a swept tip region beginning at an outward boundary of the intermediate region and extending to a tip end of the blade, the intermediate region of each blade having a sweep angle that does not decrease from the beginning to the outward boundary of the intermediate region; and

throughout the tip region the sweep angle of each blade is less than the sweep angle at the outward boundary of the intermediate region.

21. The turbomachinery of claim 20, wherein the intermediate region of each blade is swept rearward and the tip region is translated forward.

22. The turbomachinery of claim 21, wherein:

the intermediate region sweep angle of each blade increases throughout the intermediate region; and

the tip region sweep angle of each blade decreases throughout the tip region.

23. The turbomachinery of claim 22, wherein the leading edge of each blade has an inner region beginning at a root end of the blade and extending to an inward boundary of the intermediate region, the inner region being swept forward.

24. The turbomachinery of claim 20, wherein the intermediate region of each blade is swept forward and the tip region is translated rearward.

25. The turbomachinery of claim 24, wherein the tip region sweep angle of each blade decreases throughout the tip region.

26. The turbomachinery of claim 25, wherein the intermediate region sweep angle of each blade increases throughout the intermediate region.

27. A gas turbine engine fan comprising a plurality of identical blades, each blade being mounted for rotation within a case circumscribing the blades and having an inner wall forming an outer boundary for a working medium gas flowing through passages formed by neighboring blades, wherein:

each blade has a configuration enabling the fan to rotate at speeds providing supersonic working medium gas velocities in the vicinity of the passages proximate to the case;

each blade has a leading edge with an inner region, an intermediate region and a tip region, the inner region beginning at a root end of

the blade and extending to an inward boundary of the intermediate region, and the tip region extending from an outward boundary of the intermediate region to a tip end of the blade; and

the inner region is swept forward, the intermediate region is swept rearward at a sweep angle that does not decrease, and the tip region is translated forward from the outward boundary of the intermediate region.

28. The gas turbine engine fan of claim 27, wherein the tip region maintains a rearward sweep throughout the tip region.

29. The gas turbine engine fan of claim 27, wherein:

the intermediate region sweep angle of each blade increases throughout the intermediate region; and

the tip region of each blade is swept at a sweep angle that decreases throughout the tip region.

30. A blade for a gas turbine engine rotatable within a case at speeds providing supersonic flow over at least a portion of the blade, wherein the blade leading edge has a rear swept middle region ending at a tip region that is translated forward from the end of the middle region.

31. The blade of claim 30, wherein the tip region maintains a rearward sweep throughout the tip region.

32. The blade of claim 30, wherein the leading edge has a forward swept inner region.

33. The blade of claim 32, wherein the sweep angle of the middle region increases throughout the middle region.

34. The blade of claim 33, wherein throughout the tip region the sweep angle is less than the sweep angle at the end of the middle region.

35. The blade of claim 34, wherein the sweep angle of the tip region decreases from the end of the middle region to a tip end of the blade.

36. A blade for a gas turbine engine rotatable within a case at speeds providing supersonic flow over at least a portion of the blade, wherein the blade leading edge has a forward swept middle region ending at a tip region that is translated rearward from the end of the middle region.

37. The blade of claim 36, wherein the tip region maintains a forward sweep throughout the tip region.

38. The blade of claim 36, wherein the leading edge has a rear swept inner region.

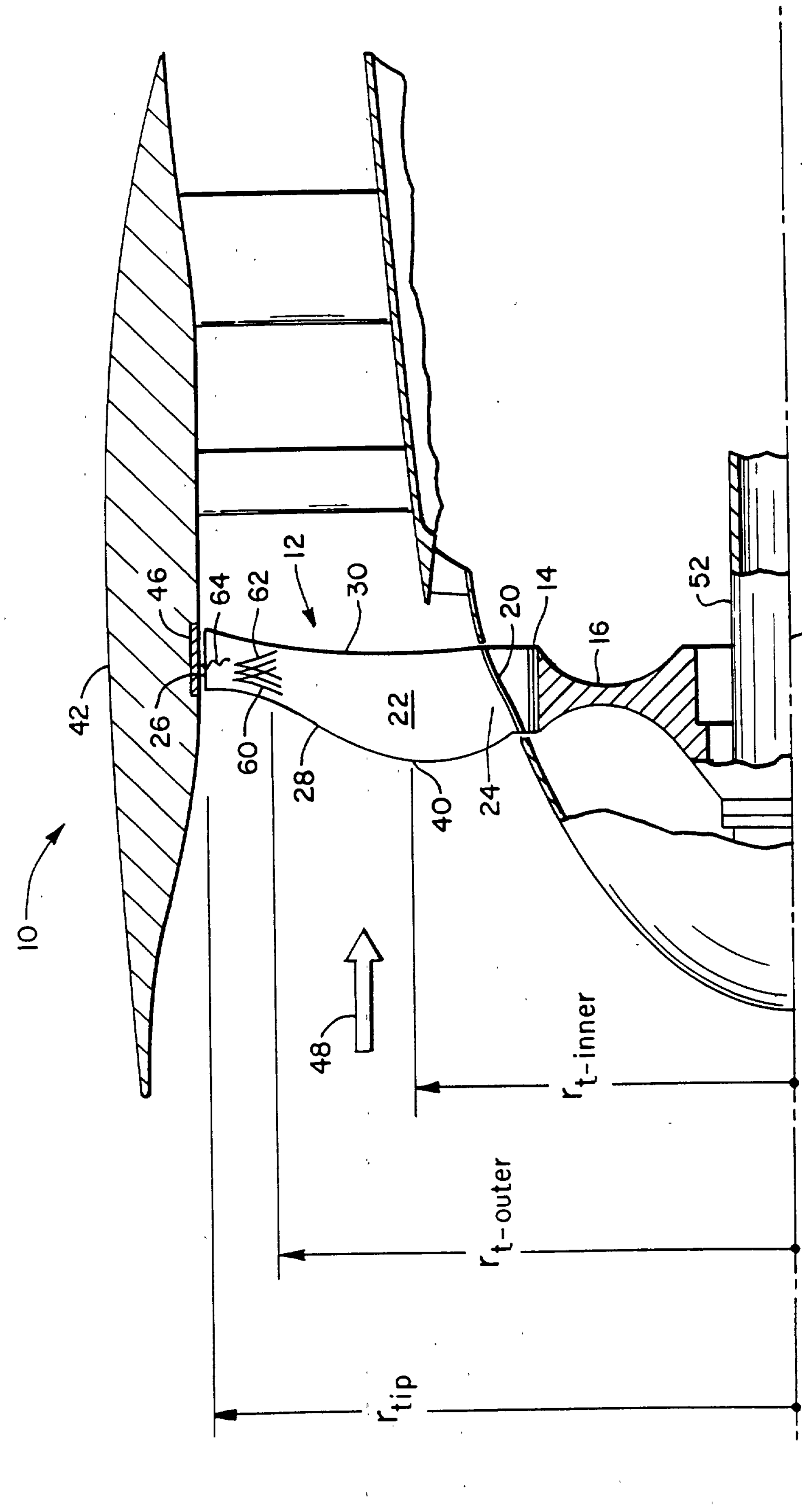
39. The blade of claim 38, wherein the sweep angle of the middle region increases throughout the middle region.

40. The blade of claim 39, wherein throughout the tip region the sweep angle is less than the sweep angle at the end of the middle region.

41. The blade of claim 40, wherein the sweep angle of the tip region decreases from the end of the middle region to a tip end of the blade.

ABSTRACT

A swept turbomachinery blade for use in a cascade of such blades is disclosed. The blade (12) has an airfoil (22) uniquely swept so that an endwall shock (64) of limited radial extent and a passage shock (66) are coincident and a working medium (48) flowing through interblade passages (50) is subjected to a single coincident shock rather than the individual shocks. In one embodiment of the invention the forwardmost extremity of the airfoil defines an inner transition point (40) located at an inner transition radius r_t-inner. The sweep angle of the airfoil is nondecreasing with increasing radius from the inner transition radius to an outer transition radius $r_{t\text{-}outer}$ radially inward of the airfoil tip (26), and is nonincreasing with increasing radius between the outer transition radius and the airfoil tip.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60
22
28
40
24
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52
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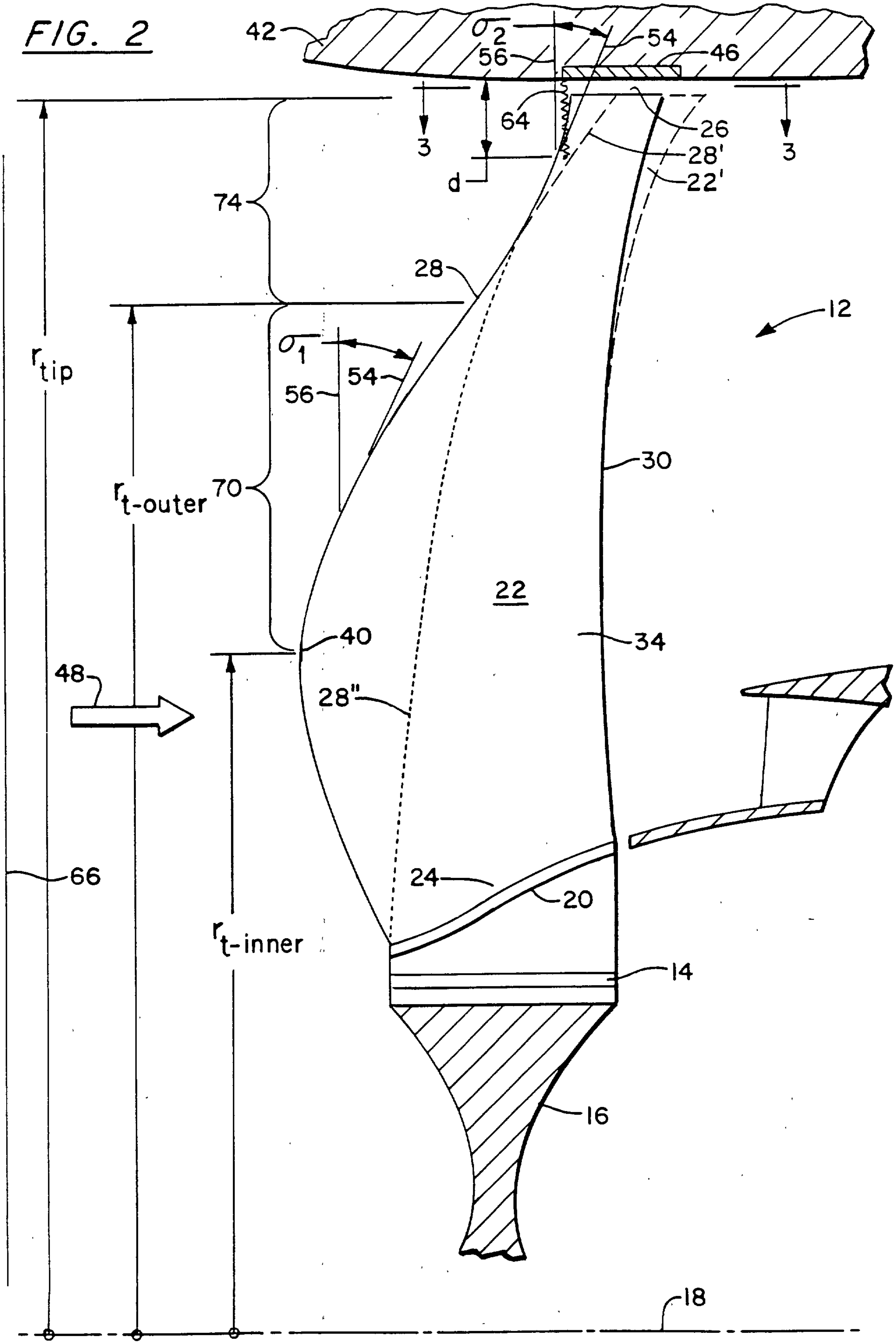
$r_{t\text{-inner}}$
$r_{t\text{-outer}}$
r_{tip}

FIG. 1

FIG. 2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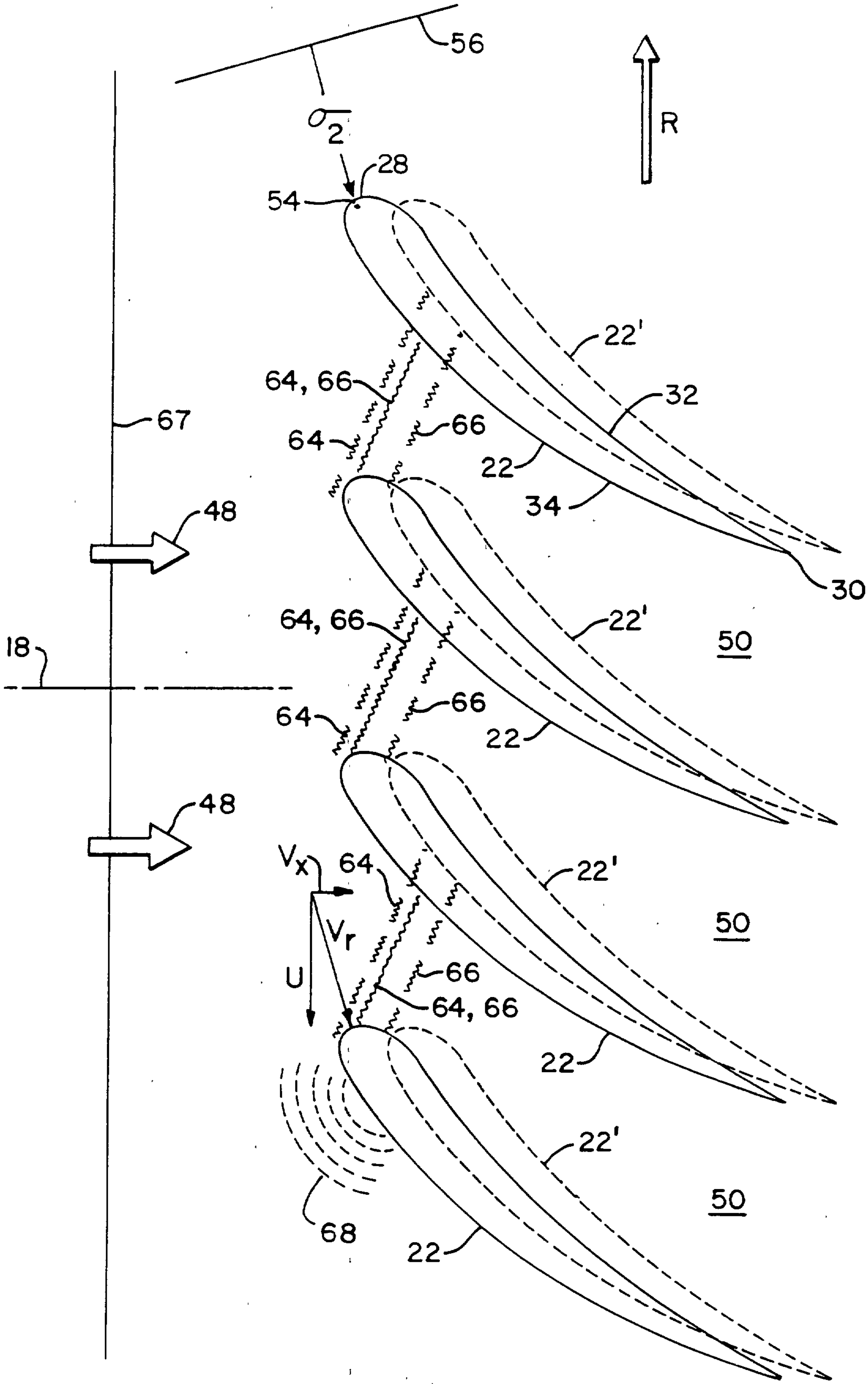

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66
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FIG. 3

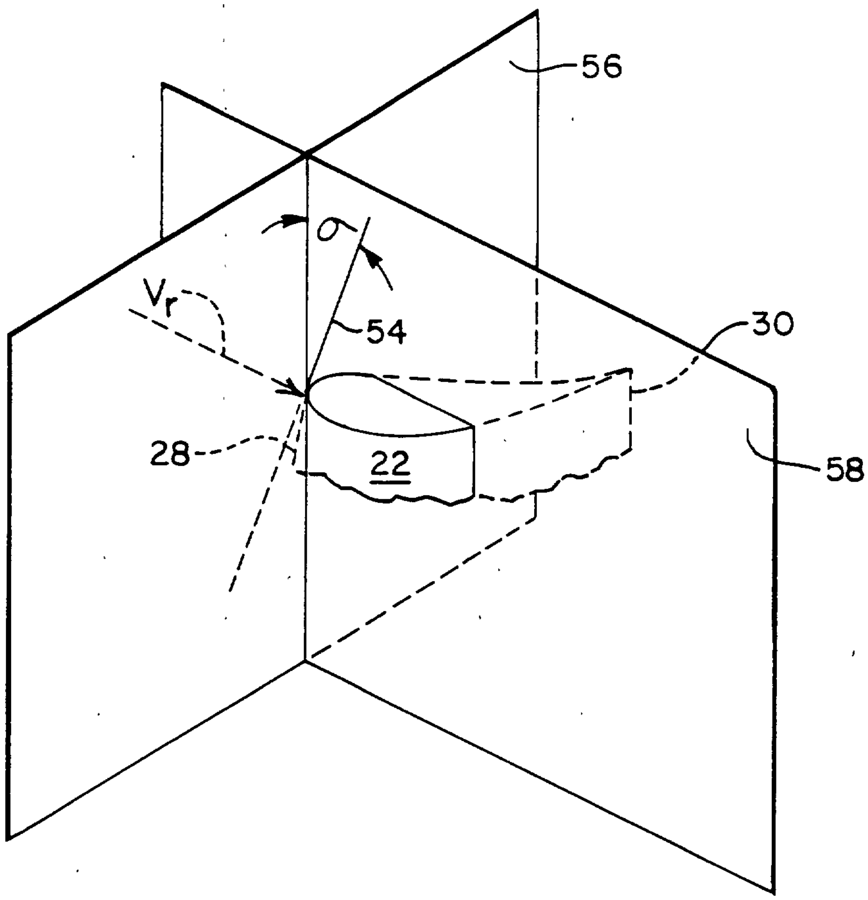

FIG. 4

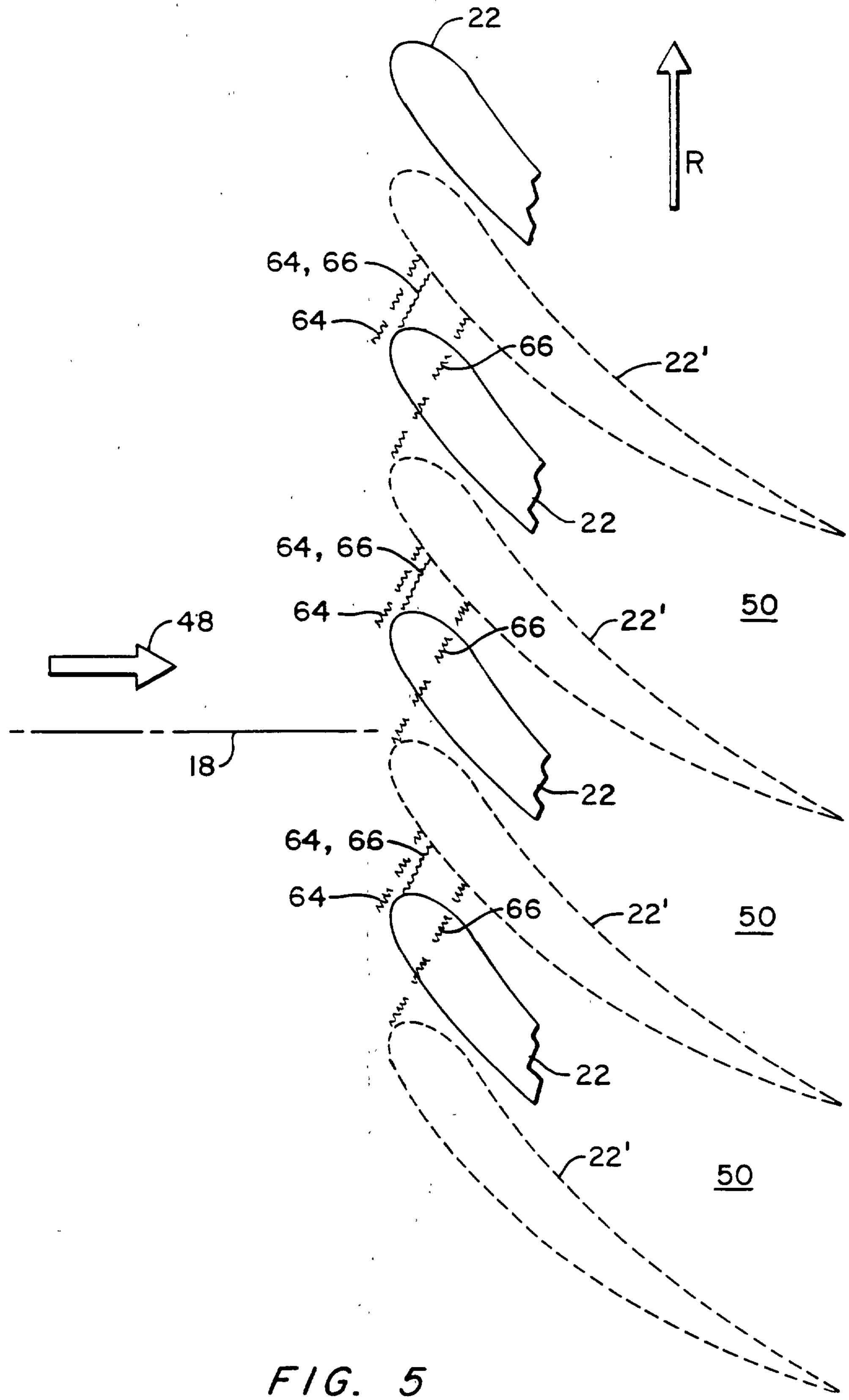
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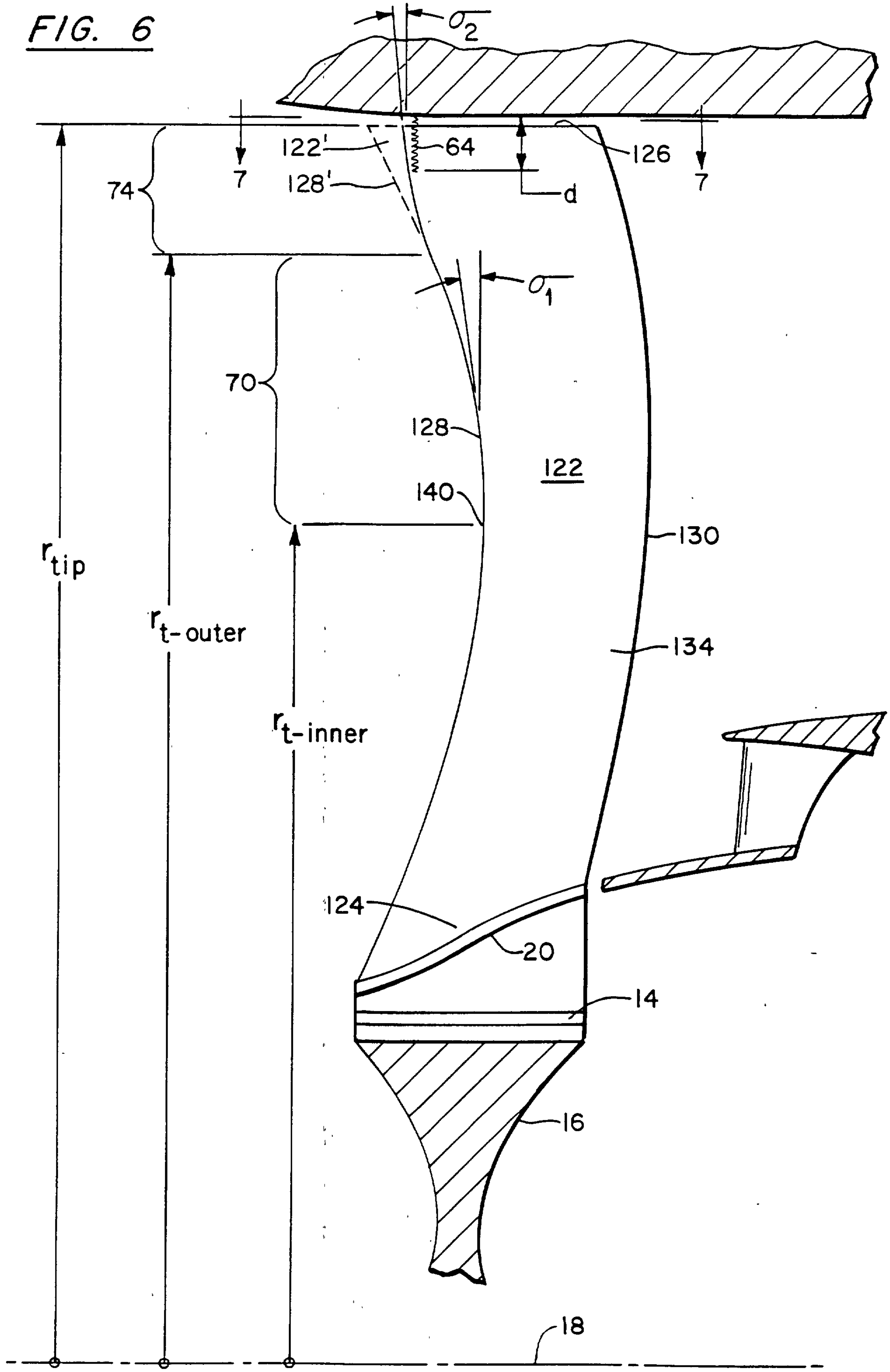
FIG. 5

FIG. 6



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r t-outer
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r t-inner
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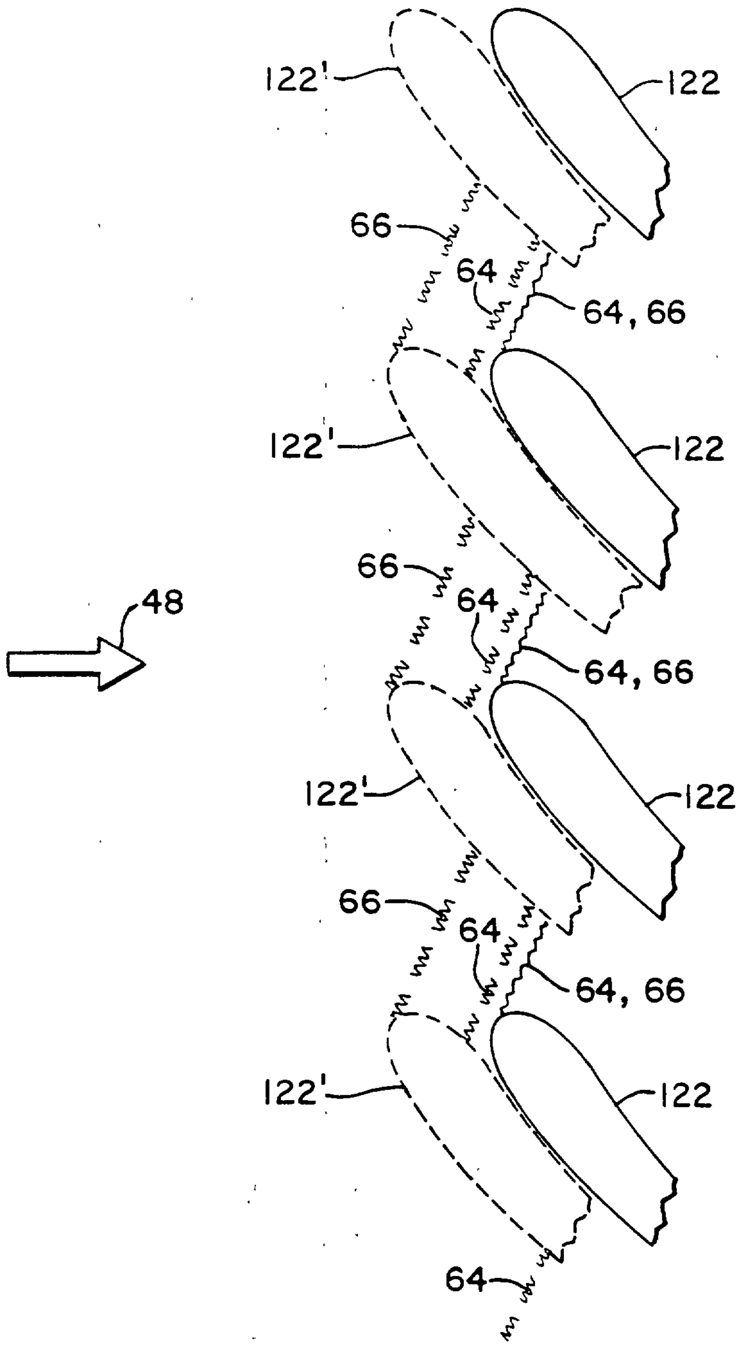
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122'
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64

FIG. 7

EXHIBIT C

INFORMATION DISCLOSURE CITATION

(Sheet 1 of 2)

OIPE JC79 JAN 0 5 2000 PATENT & TRADEMARK OFFICE

Atty. Docket No.: 3600.100	Application No.: 09/343,736
Applicants: David A. Spear et al.	
Filing Date: June 30, 1999	Group: 3745

Duplicate IDS cv 10/6/00

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U.S. PATENT DOCUMENTS

*Examiner Initial		Document Number							Date	Name	Class	Subclass	Filing Date (if appropriate)
		1	9	6	4	5	2	5	6/26/34	McMahan	170	159	
		2	1	5	4	3	1	3	4/11/39	McMahan	230	274	
		2	6	2	8	7	6	8	2/17/53	Kantrowitz			
		2	6	8	9	6	8	1	9/21/54	Sabatiuk			
		2	9	1	5	2	3	8	12/1/59	Szydlowski	230	134	
		2	9	3	4	2	5	9	4/26/60	Hausmann	415	181	
		2	9	3	5	2	4	6	5/3/60	Roy	415	181	
		3	4	1	6	7	2	5	12/17/68	Bohanon	230	259	
		3	4	4	4	8	1	7	5/20/69	Caldwell	103	88	
		3	6	9	2	4	2	5	9/19/72	Erwin	415	181	
		3	9	8	9	4	0	6	11/2/76	Bliss	415	1	
		4	0	1	2	1	6	5	3/15/77	Kraig	415	145	
		4	0	1	2	1	7	2	3/15/77	Schwaar et al.	416	228	
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	FR	2	4	5	9	3	8	7	1/9/81	France			Yes
	EP	0	2	6	6	2	9	8	5/4/88	Europe			
	SU	1	5	2	8	9	6	5	12/15/89	U.S.S.R.			No
	WO	9	1	0	7	5	9	3	5/30/91	PCT			
	EP	0	7	7	4	5	6	7	5/21/97	Europe			
	EP	0	8	0	1	2	3	0	10/15/97	Europe			

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		Cheatham et al., "Parametric Blade Study," November 1989, Report No. WRDC-TR-89-2121, Aero Propulsion and Power Laboratory, Wright Aeronautical Research & Development Center, Wright-Patterson Air Force Base, Ohio

EXAMINER	DATE CONSIDERED

*Examiner: Initial if reference considered, whether or not citation is in conformance with MPEP 609. Draw line through citation if not in conformance and not considered. Include copy of this form with next communication.

Form PTO-1449

INFORMATION DISCLOSURE CITATION

(Sheet 2 of 2)

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Filing Date: June 30, 1999	Group: 3745

U.S. PATENT DOCUMENTS

*Examiner Initial		Document Number	Date	Name	Class	Subclass	Filing Date (if appropriate)
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		4 7 2 6 7 3 7	2/23/88	Weingold et al.	416	223A	
		4 7 3 7 0 7 7	4/12/88	Vera	416	242	
		4 7 8 4 5 7 5	11/15/88	Nelson et al.	416	226	
		5 0 6 4 3 4 5	11/12/91	Kimball	416	169A	
		5 1 1 2 1 9 2	5/12/92	Weetman	416	201A	
		5 1 6 7 4 8 9	12/1/92	Wadia et al.	415	182.1	
		5 5 8 4 6 6 1	12/17/96	Brooks	416	238	

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cv
10/6/00

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FOREIGN PATENT DOCUMENTS

		Document Number	Date	Country	Class	Subclass	Translation (Yes/No)

OTHER DOCUMENTS (Including Author, Title, Date, Pertinent Page, Etc.)

		European Search Report, dated February 25, 1998, in EP 774,567
		European Patent Office Official Action, dated September 24, 1998, in EP 774,567

EXAMINER	DATE CONSIDERED

*Examiner: Initial if reference considered, whether or not citation is in conformance with MPEP 609. Draw line through citation if not in conformance and not considered. Include copy of this form with next communication.

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